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HYPERSONIC FLOWS AS RELATED TO THE NATIONAL AEROSPACE PLANE

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16. Abstract

The object of Cooperative Agreement NCC2-452 was to identify, develop, and document reliable turbulence models for incorporation into CFD codes, which would then subsequently be incorporated into numerical design procedures for the NASP and any other hypersonic vehicles. In a two-pronged effort, consisting of an experimental and a theoretical approach, several key features of flows over complex vehicles were identified, and test bodies were designed which were composed of simple geometric shapes over which these flow features were measured. The experiments were conducted in the 3.5' Hypersonic Wind Tunnel at NASA Ames Research Center, at nominal Mach numbers from 7 to 8.3 and Re/m from 4.9×10^6 to 5.8×10^6. Boundary layers approaching the interaction region were 2.5 to 3.7 cm thick. Surface and flow field measurements were conducted, and the initial boundary conditions were experimentally documented.

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To design realistic aerodynamic vehicles to fly in the hypersonic flow regime, it is of primary importance to be able to predict, with reasonable reliability, the aerodynamic characteristics of such vehicles. Extended and expensive design programs can thereby be significantly improved, and efficient designs identified and studied. Before attempting to predict the aerodynamics of the flow over a complex vehicle flying at angle of attack, one should be able to reliably predict basic flow properties, such as surface pressures, heat transfer distributions, skin friction lines, extent of separation (if any), flow direction, etc., on simple generic shapes. Without the ability to verify computations by experiment on a simple generic body, attempting to predict the flow field over a complex body would be unproductive. This cooperative agreement was initiated in order to investigate such generic hypersonic flows, both experimentally and computationally, particularly in relation to the National Aerospace Plane (NASP).

The thrust of this research was two fold. Primarily, since there was a lack of reliable well documented basic hypersonic 2 - D and 3 - D hypersonic experiments (as documented by Settles and Dodson in references 1, 2, and 3), there was a compelling reason to provide such an experimental database. Secondly, there was a critical need to provide the necessary experimental data to enable the computational fluid dynamists (cfd) to validate, invalidate, improve existing and develop new turbulence models. Only in this manner, in which the basic physical aspects of hypersonic flow, including large boundary layers interacting with shock waves, can be isolated and carefully studied both experimentally and computationally, would we be able to eventually reach the desired goal of predicting the aerodynamics of a realistic complex hypersonic body, such as the NASP. Cooperative agreement # NCC2 - 452 was initiated to support this program. The object of this program was to identify, develop, and document reliable turbulence models which could be incorporated into CFD codes, which would then subsequently be incorporated into numerical design procedures for the NASP and any other hypersonic vehicles. In order to accomplish this objective, a two pronged effort, consisting of both an experimental and theoretical approach was made. Several key features of flows over such complex vehicles were identified, and test bodies designed which were composed of simple geometric shapes over which these flow features can be measured.

EXPERIMENTAL INVESTIGATIONS

The experiments were run in the 3.5' Hypersonic Wind Tunnel at NASA Ames Research Center. They were run at nominal Mach numbers from 7 to 8.3 at Re/m from 4.9×10^6 to 5.8×10^6. Boundary layers approaching the interaction region were relatively large (approximately 2.5 to 3.7 cm thick), which allowed detailed flow field surveys to be easily made. In most of the experiments, both surface (p_w, q_w, skin friction lines) and flow field measurements (P_T, T_T, flow angle) were made, and the initial boundary conditions were experimentally documented.

The rationale driving this cooperative agreement was two fold. Primarily, an experimental investigation of a complex flow similar to one encountered in real life but reduced to its essential modular elements could provide a physical insight into the flow field. Secondly, these experiments were performed to provide a reliable data base against which various turbulence models could be validated (or as was more often the case, invalidated). To accomplish this, both surface and flow field measurements were made, initial boundary layer conditions were measured, and an error analysis was made. The resulting data, besides being analyzed and presented in archival literature (AIAA journals, for example) and technical meetings, were tabulated and published as NASA TM's, with the data available on floppy disks. Because of the above approach, the experiments were per se of high quality, and all of them (with two minor exceptions) passed the stringent requirements of the comprehensive Settles - Dodson database studies discussed in reference 1 , 2, and 3.

The experiments discussed below were performed by the principal investigator, Marvin Kussoy, under the subject cooperative agreement.

M = 7 hypersonic cone and wedge flows

The test bed employed in the present study consisted of a cone / ogive / cylinder at zero angle of attack. Attached to the cylinder were a series of axisymmetric flares or symmetric sharp fins. (figure 1.) Both the flare and fin angles were varied, producing shock waves of various strengths, and resulting in both attached and separated flow fields. Detailed boundary - layer surveys have verified a fully developed turbulent layer on the cylinder ahead of the interaction region. The resulting flows were axisymmetric

(with and without separation) and 3 - D (with separation). These particular flows were chosen because , as stated above, they were relatively simple, but exhibited the same basic characteristics as complex hypersonic vehicles. Surface pressures and heat transfer rates were measured on both configurations. In addition, flow field surveys were done on the flare configuration.

The experimental results were reported in reference 4 .

M = 8.2 2D impinging shock and 3D wedge flows

A flat plate test bed was designed for this experimental series. It is shown in figure 2. This test bed was of hollow modular construction, enabling both test bodies and instrumentation to be easily manipulated and changed. The full length of the 3.5' test section was utilized in the test bed design - and because of this, there was a equilibrium fully developed hypersonic boundary - layer 3.7 cm thick over the rear portion of the test bed - ideal with which to study shock - wave turbulent - boundary layer interactions. Two configurations were tested; the first configuration consisted of a sharp wedge the width of the test bed supported over the bed; the second a series of sharp fins attached to the plate surface. These are both illustrated in figure 2. Both the wedge and fin angles were varied, producing shock waves of various strengths. This resulted in both attached and separated flow fields for the wedge flows, and complex three - dimensional separated flow fields for the fin cases. Detailed boundary layer surveys verified a fully developed hypersonic boundary layer on the flat plate alone. These particular flows were chosen because they were relatively simple geometrically, but yet exhibited the same basic characteristics present on complex hypersonic vehicles.

For the wedge configuration, only surface conditions were measured. For the fin configuration, however, mean flow - field surveys, both in the undisturbed and interaction regimes, were taken; from them, incoming boundary - layer parameters and pitot pressure contours were obtained. These data (the first to our knowledge to be obtained at hypersonic speeds for a three - dimensional shock - wave / turbulent - boundary - layer interaction flow) have sufficient resolution and accuracy to fully document this complex three - dimensional flow field. In addition, they are meant to be used as a data base with which to validate existing of future computational models of these hypersonic flows.

These results were analyzed and reported at technical meetings (references 5, and 6), published in an archival journal (reference 7), and also published in a convenient database format as an NASA TM (reference 8).

M = 8.3 crossing shock flow

Several key elements of a generic hypersonic inlet were identified - i.e. a thick turbulent boundary layer approaching two vertical fins, a crossing shock pattern, vortices, large pressure gradients, and separation zones. The particular configuration was chosen because it was relatively simple, yet exhibited the same basic characteristics present in generic inlets that would be present on complex hypersonic vehicles. The experiment was designed to generate flows with varying degrees of pressure gradient, boundary - layer separation, and turning angle The test bodies for this experimental series consisted of two symmetrical sharp fins attached to the flat plate test bed as shown in figure 3.

Streamwise and transverse surface pressure and heat transfer distributions, as well as flow - field surveys which measured pitot pressures and yaw angles in the interaction regime were measured. One important result deserves special attention. That is, the persistence of an extensive low pressure region far downstream of the fin leading edges. This low pressure region implied that the generic inlet tested here, in which a thick turbulent boundary - layer approaches two vertical compression surfaces, would not be a particularly efficient pressure increasing device.

These experimental results were presented at a technical meeting (ref. 9), published in an archival journal (ref. 10), and also presented as a database (ref. 11) for validating present or future turbulence models and computer codes.

Two additional and logical extensions of this test program were planned for the future. One was the crossing shock flow at a positive (not zero) yaw angle on the flat plate test bed. The other was the crossing shock flow with a " roof " or cowl, to provide a compression shock to the incoming flow. (This complex 3 - D flow has, in fact, been computed.) These tests were meant to more closely simulate the complex flow and conditions a realistic hypersonic inlet would experience. The last year was spent planning this experiment, designing the test bodies and probes. However an avoidable disaster during the 3.5 ' hypersonic wind tunnel major rehab, coupled with budgetary conditions combined to terminate these future tests.

THEORETICAL INVESTIGATIONS

Turbulence Compressibility Corrections

In addition to the experimental investigations discussed above, in which computations were specifically made to try to predict the experimental results using existing computer codes and turbulent models, George Huang, operating under the same cooperative agreement, instituted a series of investigations of the above as well as other experimental work, with the object of modifying and improving the pertinent computer codes being used for these predictions

Turbulence models were corrected, and the experiments selected to substantiate these model corrections were based on the experimental database presented by Settles and Dodson in references 1, 2, and 3. The configurations of these experiments are shown in figure 4. All calculations were made using a code developed in 1992, and discussed in references 12 to 18. The inlet flow conditions were obtained by calculating a flow over the flat plate until the experimental displacement is matched. The y^+ value of the first grid point was maintained to be less than 0.5 and the grid mesh was expanded from wall to the free stream at a constant rate, which was determined to ensure that at least 60 - 80 grids were inside the boundary layer. Test runs were performed with fewer grids inside the boundary layer (40 grids) and no significant differences of the solutions have been observed.

Case 1. M = 7.05, Hypersonic Cylinder - Flare

The case was a Mach 7 flow over an axi - symmetric flare investigated experimentally by Kussoy and Horstman (reference 4). The surface is cooled, and the wall to adiabatic wall temperature ratio was 0.4. The solution was obtained with a 141 by 140 mesh with 60 - 80 points inside the boundary layer. In figure 5, the predicted surface pressure and heat transfer are compared with experimental data, normalized by their corresponding experimental values at the inflow, for a flare angle of 35^0 . As can be seen from the figure, the baseline k - epsilon and the k - omega models under predict the extent of flow separation and overpredict the heat transfer rate near flow re - attachment. The modified versions of the models using the model corrections remove the above mentioned difficulties and result in successful predictions.

Case 2. M = 9.22. 2 - D Compression Corner

Another problem with similar flow features consists of a shock - wave and boundary - layer interactions induced by a 2 - D compression corner at a free stream Mach number of 9.22, investigated by Coleman and Stollery and reported in reference 19. The free - stream and surface temperatures are 64.5^0 K and 295^0 K respectively. The numerical solutions were made with a 141 by 140 mesh and with 60 - 80 points inside the boundary layer. The surface pressure and heat transfer predictions are shown in figure 6. The failures of the baseline models in predicting flow separation and surface heat transfer are clearly evident, as shown in figure 6. The modified versions of the models result in better predictions of flow separation and consequently capture the pressure peak near flow re - attachment. Furthermore, the modified models successfully predict the heat transfer rate near the flow re - attachment.

Case 3. M = 6.86, Axisymmetric Impinging Shock

The test problem consists of a cylindrical ring shock - generator used to induce shock - wave boundary layer interactions on a cone - ogive - cylinder configuration, as reported by Kussoy and Horstman in reference 20. The free - stream Mach number at the tip of the shock generation wedge was 6.86 and the temperature was 67.8^0 K. The cylinder wall temperature was fixed at 300^0 K. The computation was made with a 141 by 200 mesh and with grids being compressed both near the cylinder wall and the shock generator wedge. The comparison of experiment with the predicted surface pressure, heat transfer, and skin friction for a 15^0 ring angle is shown in figure 7. As can be seen from the figure, the baseline models fail to predict flow separation and over predict surface heat transfer rate and skin friction near flow re - attachment. Once again, the modified versions of the models correctly predict the size of flow separation and give rise to better results for the heat transfer rate and skin friction..

Aspects of the above investigation and results have been described by George Huang in references 12 - 18. In addition , further turbulence model development by him has been published in references 21 - 33. An excellent summary of this work done under this subject cooperative agreement can be found in reference 30.

Flow Separation Predictions

One of the biggest problems in aerodynamics is the correct prediction of separation of the flow from an aerodynamic body. The onset of separation determines the effectiveness of the device and the envelope of save operation. The main shortcoming in the prediction of this critical issue is the inability of turbulence models to properly account for adverse pressure gradient flow conditions. Florian Menter has set up a test base of increasingly difficult and well documented research flows consisting of the following flows:

- Free Shear Layers (far wake, mixing layer, jet)
- Flat Plate zero pressure gradient boundary layer
- Equilibrium adverse pressure gradient flows
- Moderate adverse pressure gradient boundary layer (Samuel and Joubert)
- Strong adverse pressure gradient unseparated flow (Driver)
- Strong adverse pressure gradient separated flow (Driver)
- Backward facing step flow (Driver and Seegmiller)
- NACA 4412 low speed airfoil at high angle of attack (Coles and Wadcock)
- Transonic Bump Flow at different Mach numbers (Bachalo and Johnson)
- RAE Case 10 Supercritical airfoil; M=0.75, a=3.19/2.72, Re=6.2xlO6 (Cook, McDonald and Firmin)
- MBB VA-2 Supercritical airfoil; M=0.78, a=0.9 Re=6xlO6 (Mateer, Seegmiller, Hand and Szodruch)
- NACA 64A010 Transonic airfoil; M=0.8, Re=2xlO6 (Johnson, Bachalo and Owen)
- Lockheed Wing C; M=0.9/0.883, a=4.91/4.5, Re=lxlO7 (Hinson and Burdges)
- ONEREA M6 Wing; M=0.8447, a=5.06, Re=l.lxlO7 (Schmitt and Charpin)
- 3D 60° wedge flow (Anderson and Eaton)
- 3D 60° wedge flow (Anderson and Eaton)
- Spinning cylinder flow (Driver)
- Turnaround-duct flow (Mateer and Monson)

A large number of popular turbulence models were tested against this unique data base, thereby showing the advantages and disadvantages of the different models in great detail. The following models were used in this investigation, and the results have been published in a number of papers and journal articles. They are listed here as references 34 to 47.

- Baldwin-Lomax (algebraic) model
- Johson-King (half-equation) model
- Baldwin-Barth (one-equation) model

- Spalart-Allmaras (one-equation) model
- Standard k-e (two-equation) model
- RNG k-e (two-equation) model
- Wilcox k-w (two-equation) model
- Multiscale (Reynolds stress) model

All of the above models are popular models and in widespread use in industry, many of them without having been tested properly. The studies have revealed severe shortcomings in most of these models and will help to focus the community towards the best available models.

In this extensive and thorough investigation, Menter has shown the critical importance of studying the influence of freestream values on the predicted results. Another major result found was that the k-w model was ill conditioned near the boundary layer edge, leading to a strong sensitivity with regard to free stream values. This fault can lead to severe errors in the computations. Based on these findings, Menter has developed a modification to his k - omega model that allows the correct prediction of adverse pressure gradient flows. The new model (called shear-stress transport or SST model) far outperforms all existing models of comparable complexity tested so far. The SST model has been implemented into the compressible Navier-Stokes code of the NASA Langley Research Center and has been applied successfully to two-dimensional airfoils as well as fully three-dimensional wing computations. The model has also been applied to the high-lift problem, currently studied in a joint effort of NASA, industry, and universities. The ability to correctly prediction high-lift systems is regarded as a major competitive advantage by the US airframe manufacturers. The SST model has, at this early stage of the project, already produced significantly improved results compared to the standard models used in industry today.

In addition to the work described above, Menter has also been involved in the prediction of the transition from laminar to turbulent flow. Different transition models have been tested in this study and compared against unique experimental data measured recently by two researchers of the "Modeling and Experimental Validation Branch". In another effort, Menter has supported computations of unsteady airfoil flows performed at the Ames Research Center. He was involved in the turbulence modeling effort of this study.

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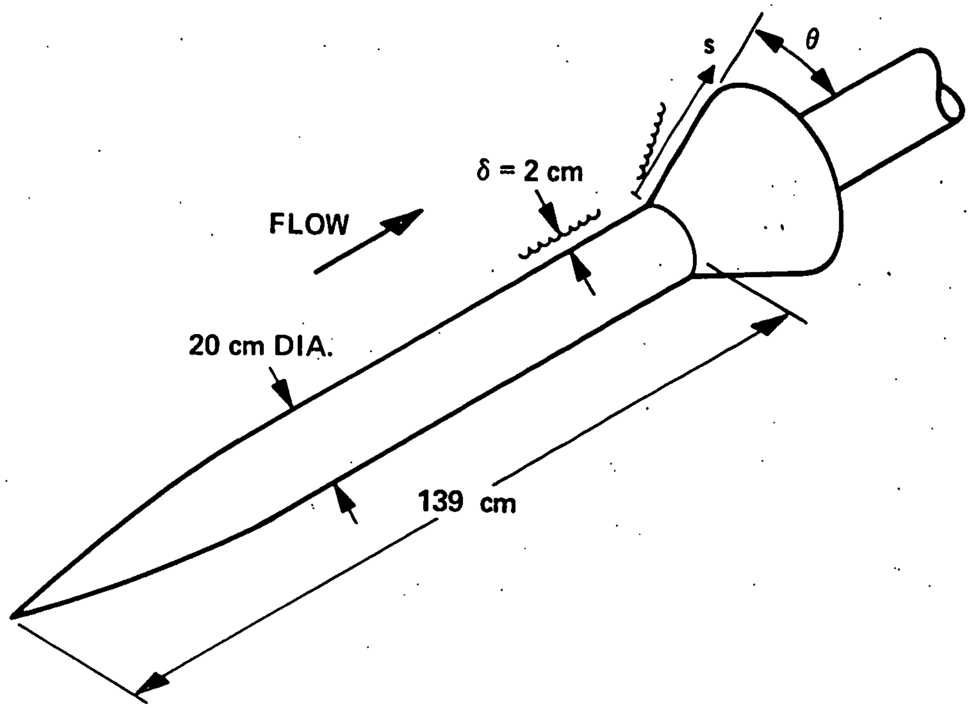
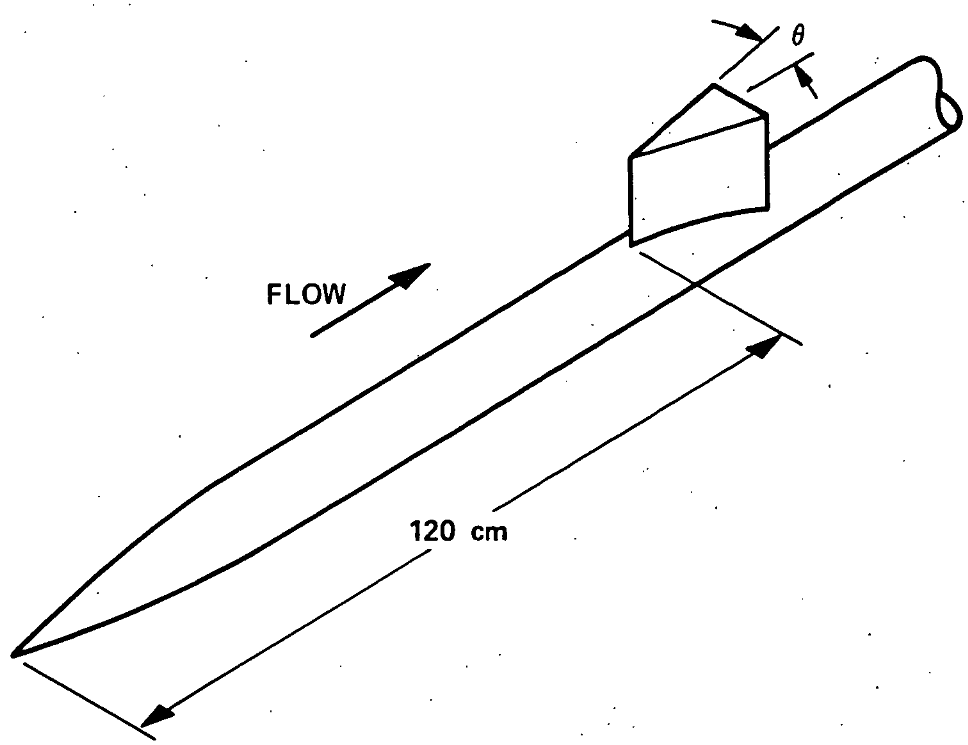


(a)



(b)

Figure 1.– Test body. (a) With flare attached. (b) With fin attached.

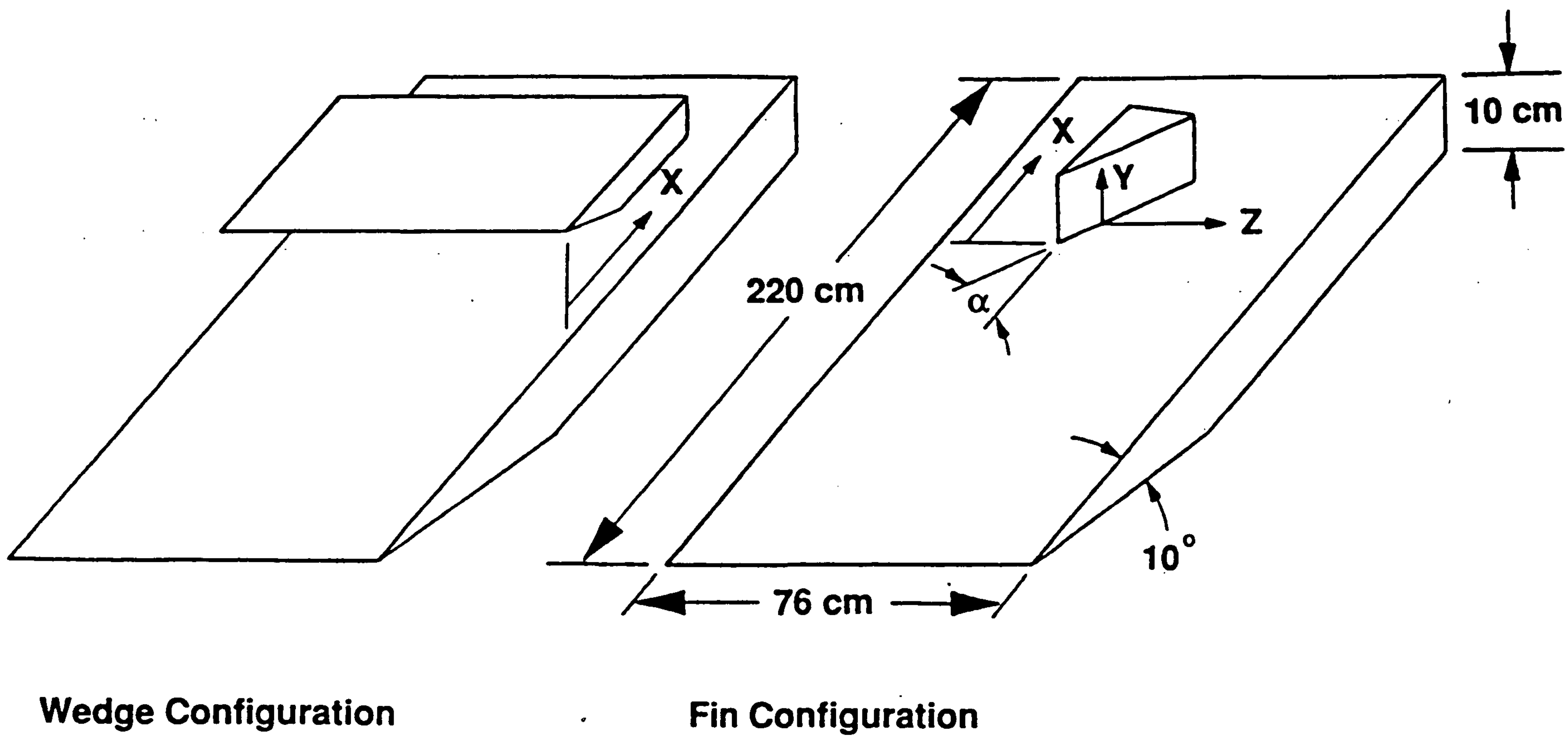


Figure 2. Test body configurations and coordinate systems used.

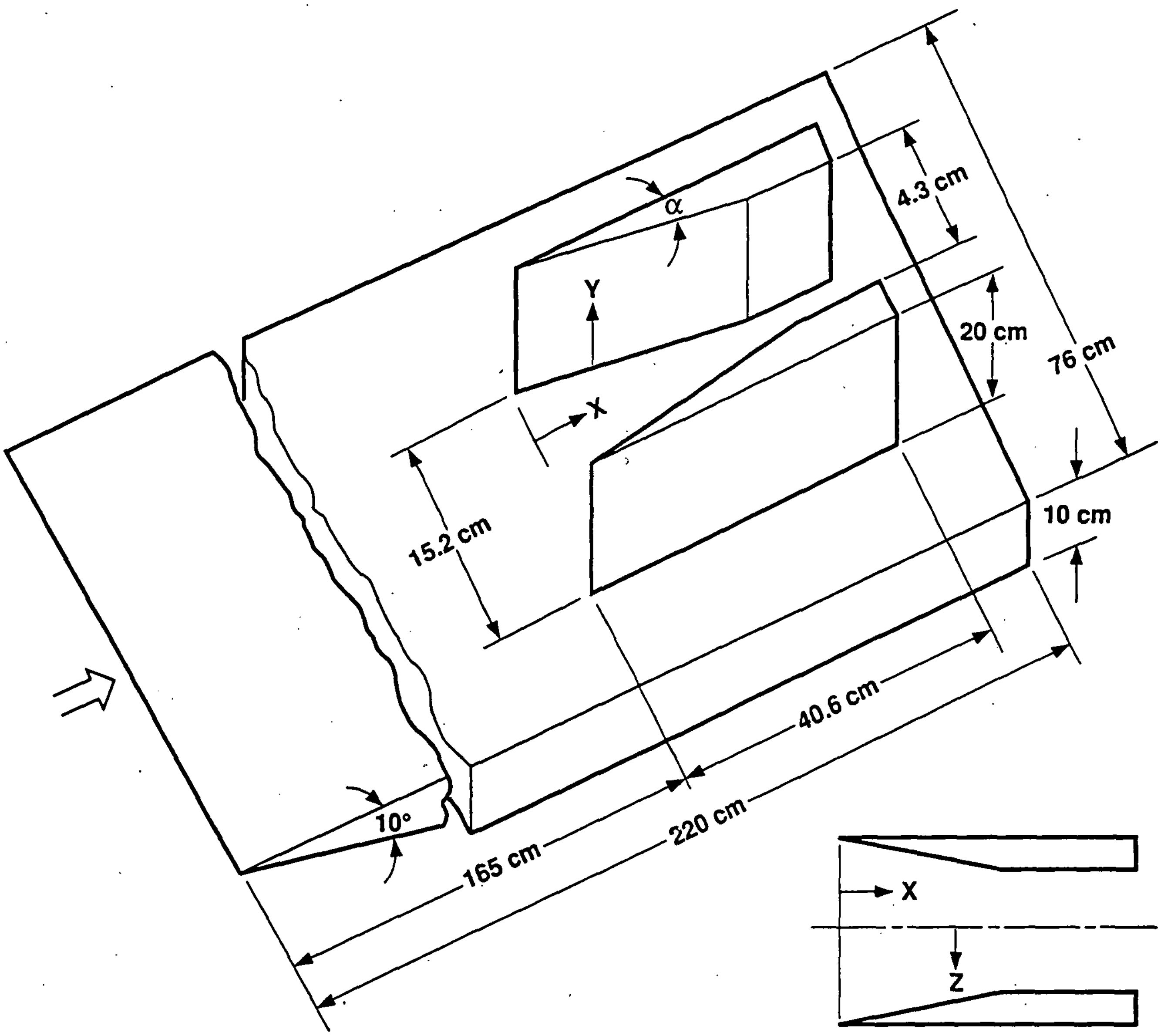


Figure 3 .Test-body configuration and coordinate system.

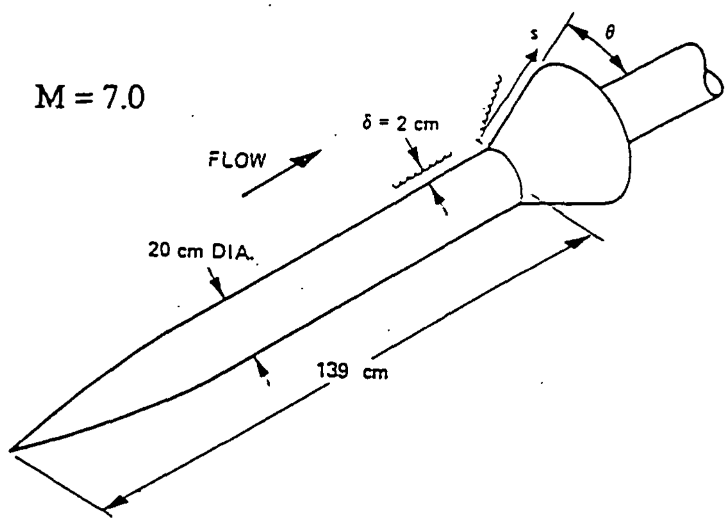


Fig. 4a. Axisymmetric compression generated with ogive-cylinder-flare configuration of Kussoy and Horstman (1989).

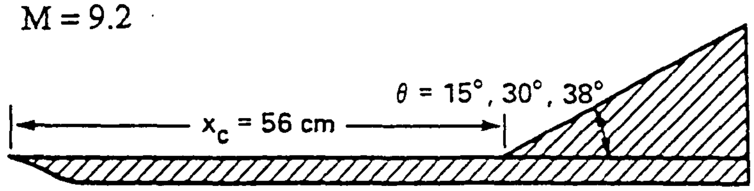


Fig. 4b. 2-D planar compression ramp configuration of Coleman and Stollery (1972).

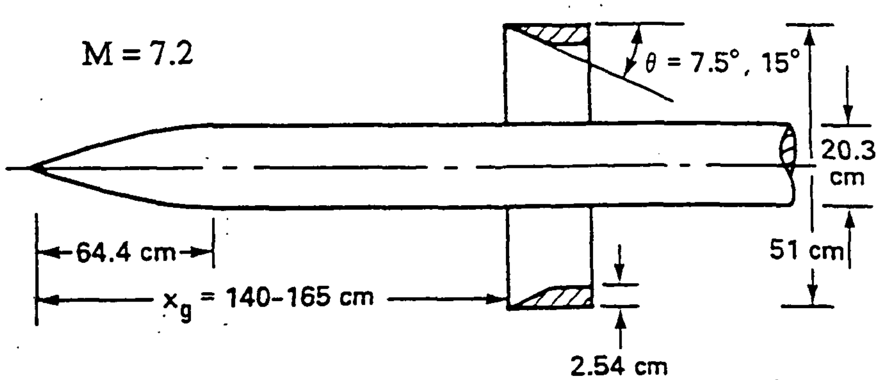
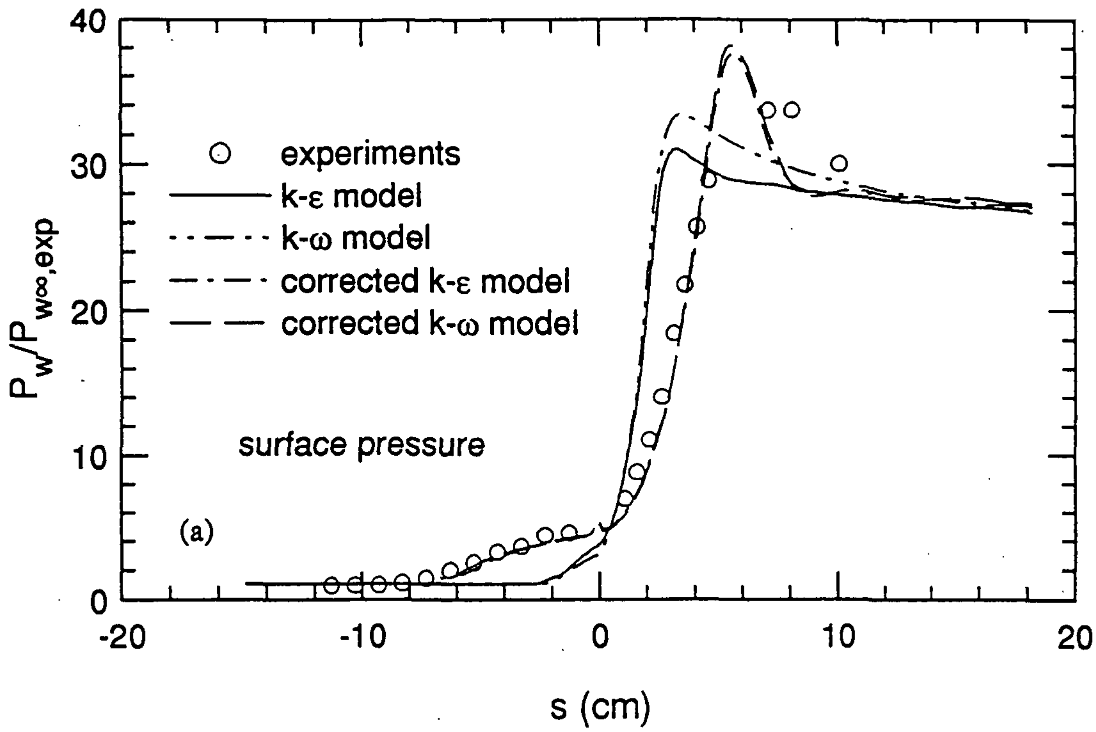


Fig. 4c. Axisymmetric impinging shock generated with a cone-ogive cylinder and an annular fin configuration of Kussoy and Horstman (1975).



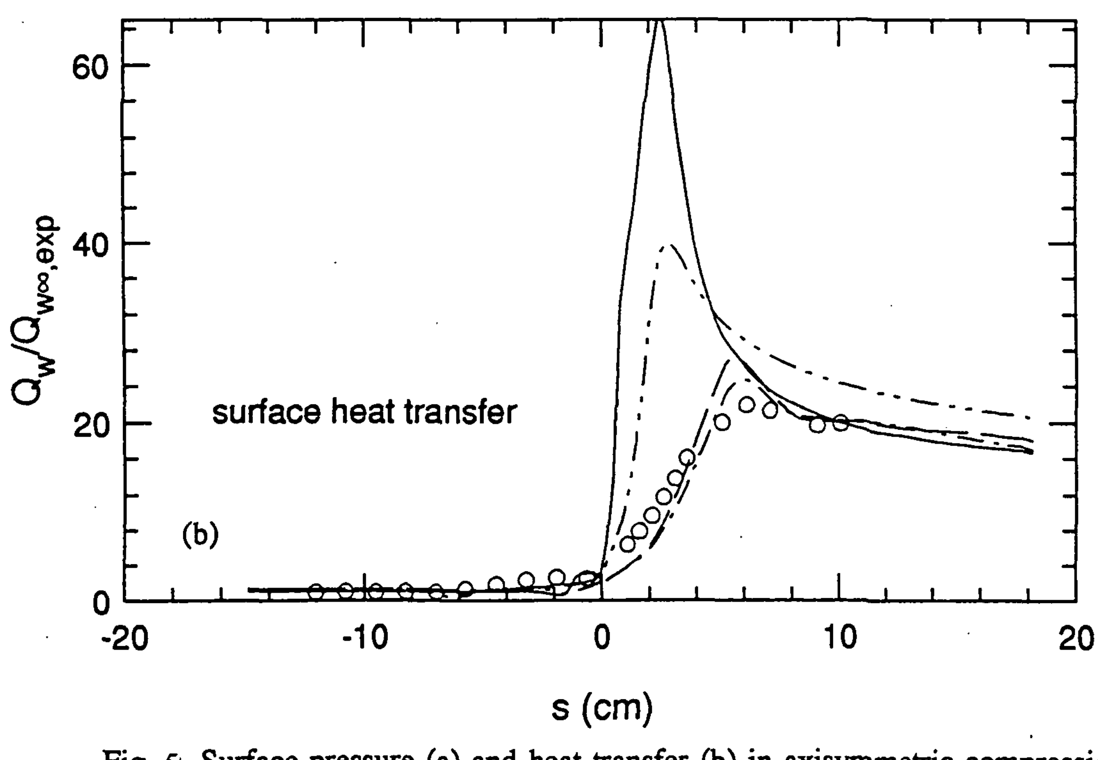
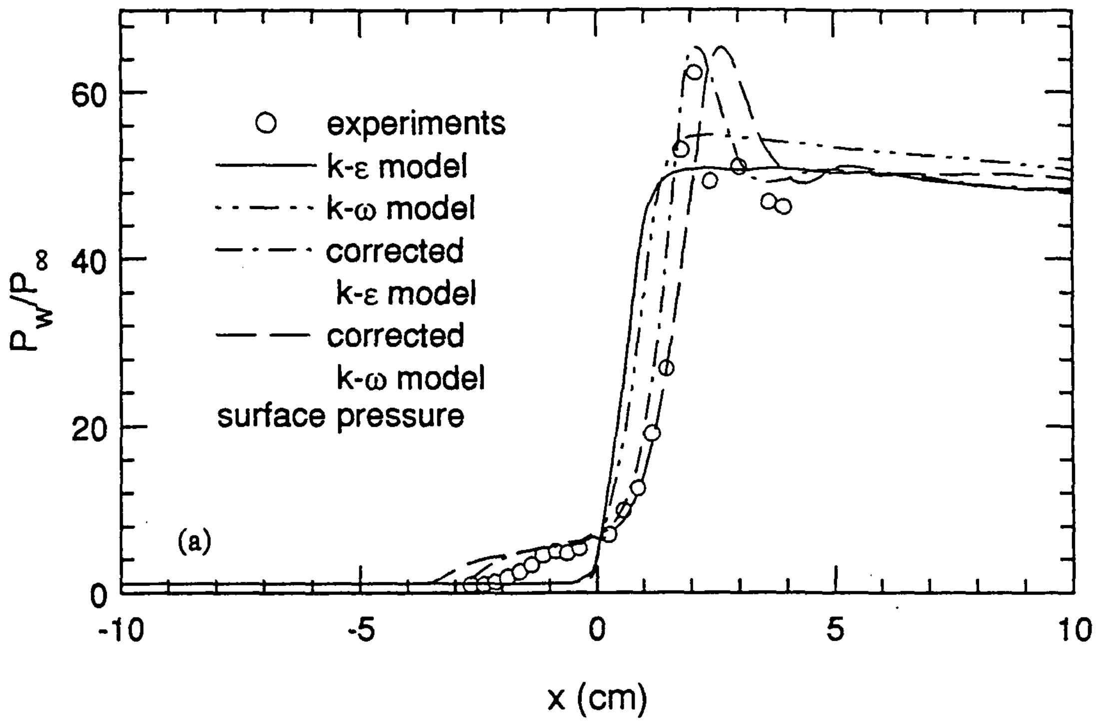


Fig. 5. Surface pressure (a) and heat transfer (b) in axisymmetric compression flow of Kussoy and Horstman (1989), geometry shown in Fig. 6a.



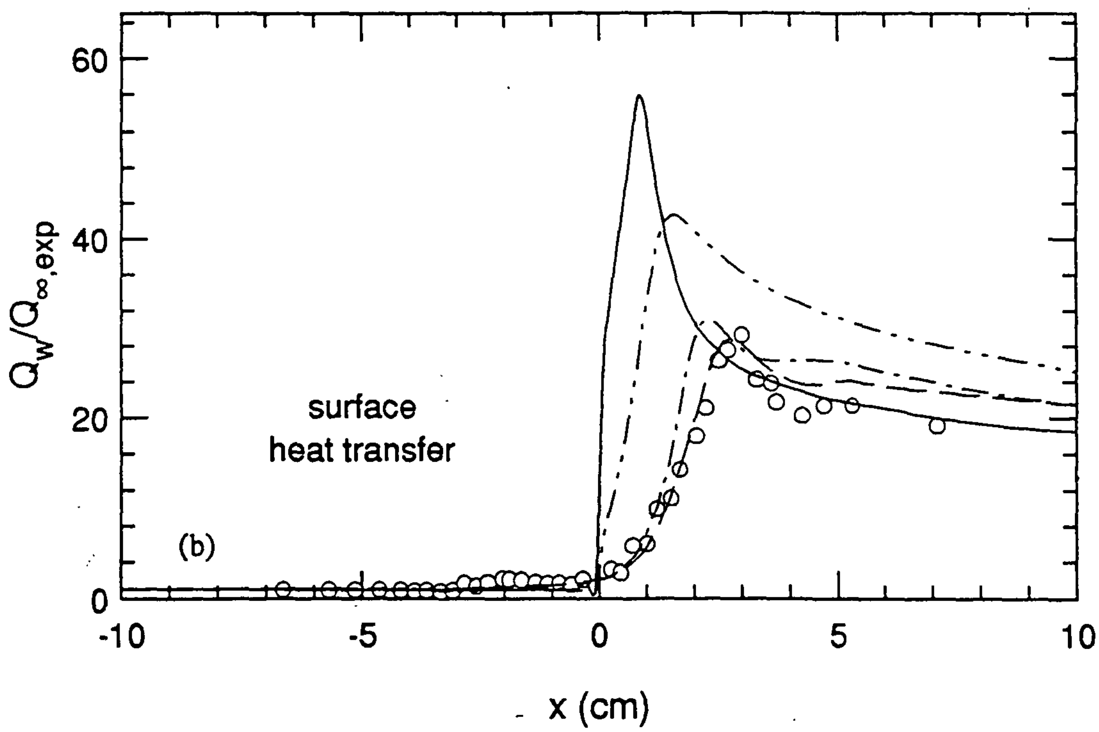


Fig. 6. Surface pressure (a) and heat transfer (b) in 2-D planar compression ramp flow of Coleman and Stollery (1972), geometry shown in Fig 6b.

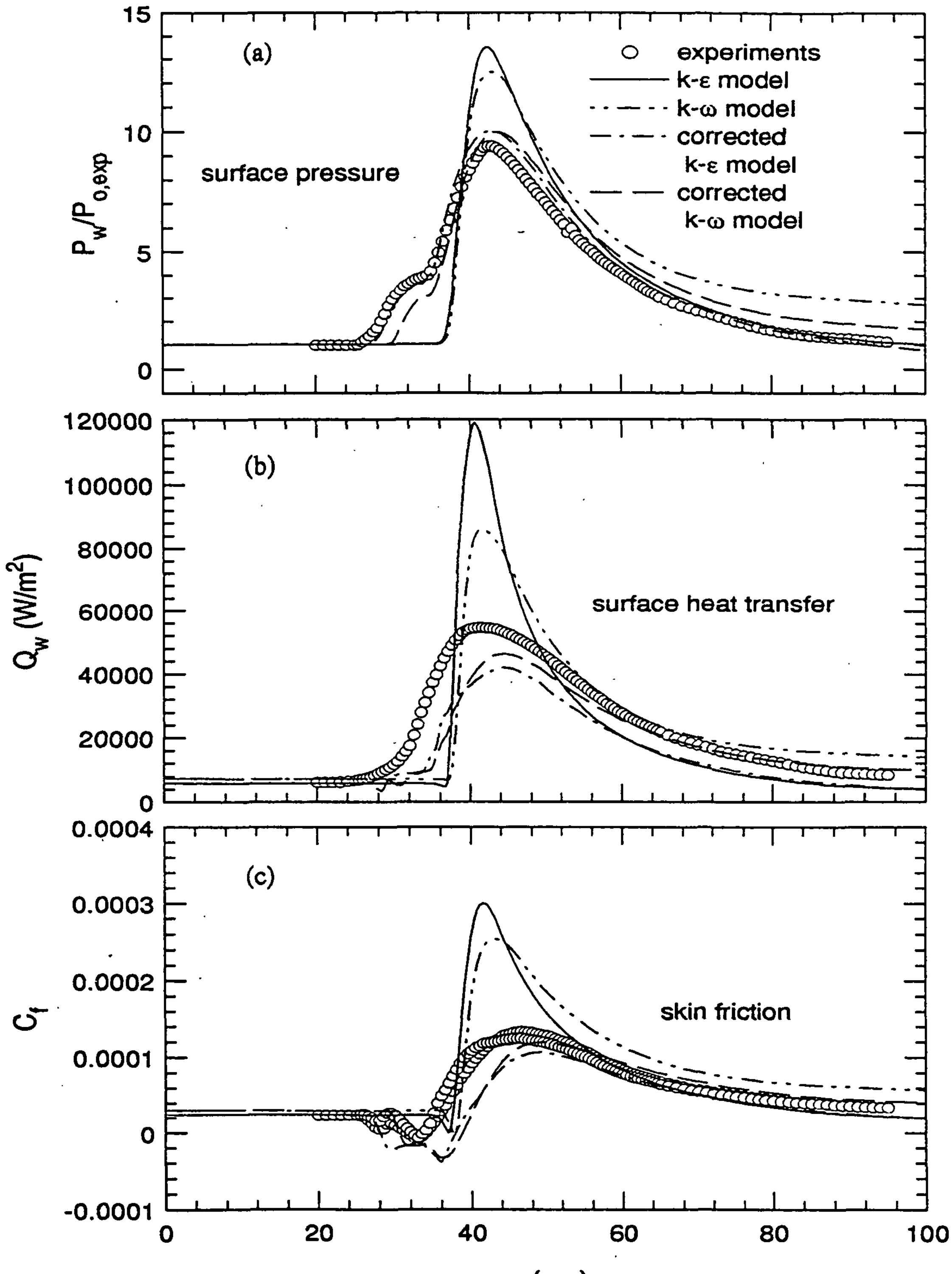


Fig. 7. Surface pressure (a), heat transfer (b) and skin friction (c) in axisymmetric impinging shock flow of Kussoy and Horstman (1975), geometry shown in Fig 6c.